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Separated Spacecraft Interferometer Concept for the New Millennium Program

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ABSTRACT

A separated spacecraft optical interferometer mission concept proposed for NASA's New Millennium Program is described. The interferometer instrument is distributed over three small spacecraft: two spacecraft serve as collectors, directing starlight toward a third spacecraft which combines the light and performs the interferometric detection. As the primary objective is technology demonstration, the optics are modest size, with a 12-cm aperture. The interferometer baseline is variable from 100 m to 1 km, providing angular resolutions from 1 to 0.1 milliarcseconds. Laser metrology is used to measure relative motions of the three spacecraft High-bandwidth corrections for stationkeeping errors are accomplished by feedforward to an optical delay line in the combiner spacecraft; low-bandwidth corrections are accomplished by spacecraft control with an electric propulsion or cold-gas system. Determination of rotation of the constellation as a whole uses a Kilometric Optical Gyro, which employs counter-propagating laser beams among the three spacecraft to measure rotation with high accuracy. The mission is deployed in a low-disturbance solar orbit to minimize the stationkeeping burden. As it is well beyond the coverage of the GPS constellation, deployment and coarse stationkeeping are monitored with a GPS-like system, with each spacecraft providing both transmit and receive ranging and attitude functions.

Key words: optical interferometry, space telescopes, spacecraft constellations, separated-spacecraft interferometry, New Millennium Program, formation flying, kilometric optical gyro

1 INTRODUCTION

Space interferometry with separated spacecraft combines the well-known advantages of space for astronomical observations with the ability to achieve extremely high angular resolutions with an easily reconfigurable system. As part of NASA's New Millennium Program, a separated spacecraft interferometer concept has been developed and is being considered as part of its deep-space mission set. The New Millennium Interferometer, NMI, is a simplified separated spacecraft interferometer that demonstrates enabling technologies while still retaining science capabilities. The technology is applicable to a separated-spacecraft implementation of the Terrestrial

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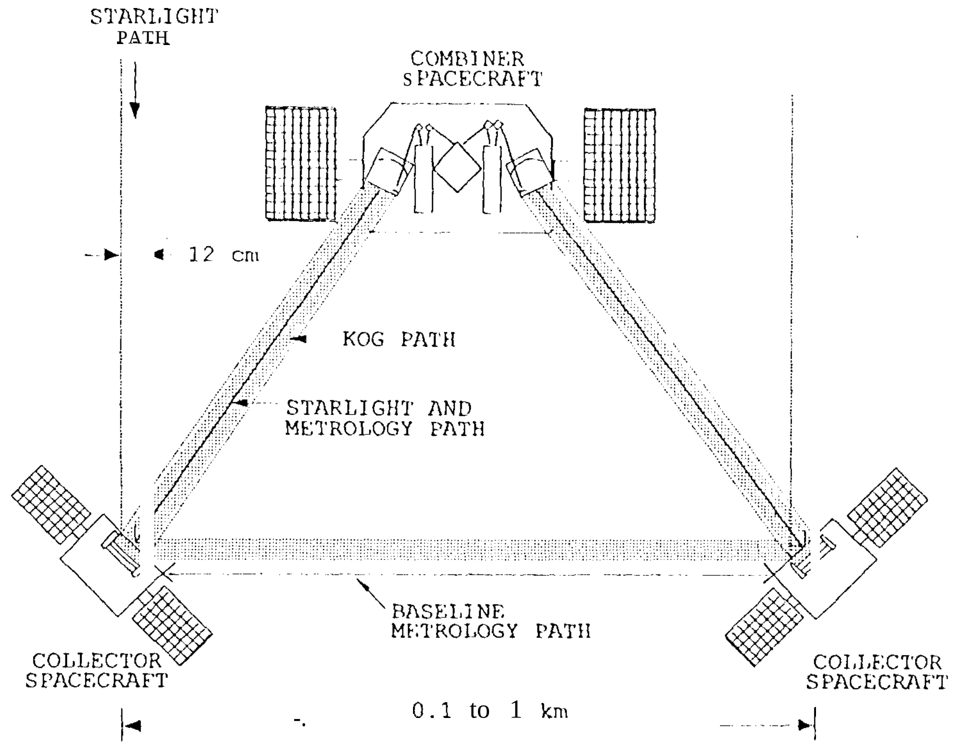

Figure 1: New Millennium Interferometer concept

Planet Finder mission, if ground measurements indicate that higher angular resolution than achievable with a connected-element system is required, and future exoplanet imaging and high resolution astrophysics missions. The formation-flying aspects of the mission have application to constellations of Earth-observing or planetary spacecraft.

Figure 1 shows the interferometer concept. Three spacecraft arranged in an equilateral triangle comprise the interferometer. Two collector spacecraft collect and relay light to a third combiner spacecraft which performs the interferometric measurements. The interferometer baseline is variable from 100 m to 1 km, providing angular resolutions in the visible of 1 to 0.1 milliarcseconds. As it is primarily a technology- demonstration mission modest 12-cm clear apertures are employed. The starlight subsystem is similar to those used in ground interferometers,[3,1] incorporating fast steering mirrors and optical delay lines for high-bandwidth tilt and pathlength control. Laser metrology among the spacecraft, in conjunction with feedforward to the optical delay lines, provides equivalent structural rigidization, similar to the approach proposed for connected-element space interferometers like the Space Interferometry Mission SIM.[4] Phasing of the interferometer uses a Kilometric Optical Gyro (KOG), a Sagnac interferometer employ ing counter-propagating laser beams among the three spacecraft. Formation flying employs an innovative sensor which uses GPS-like technology to control a small cold-gas or electric-propulsion system. The 6-month mission uses an Earth-escape orbit to minimize disturbance forces, and the nominal launch vehicle is a Delta-Lite. These aspects are discussed below.

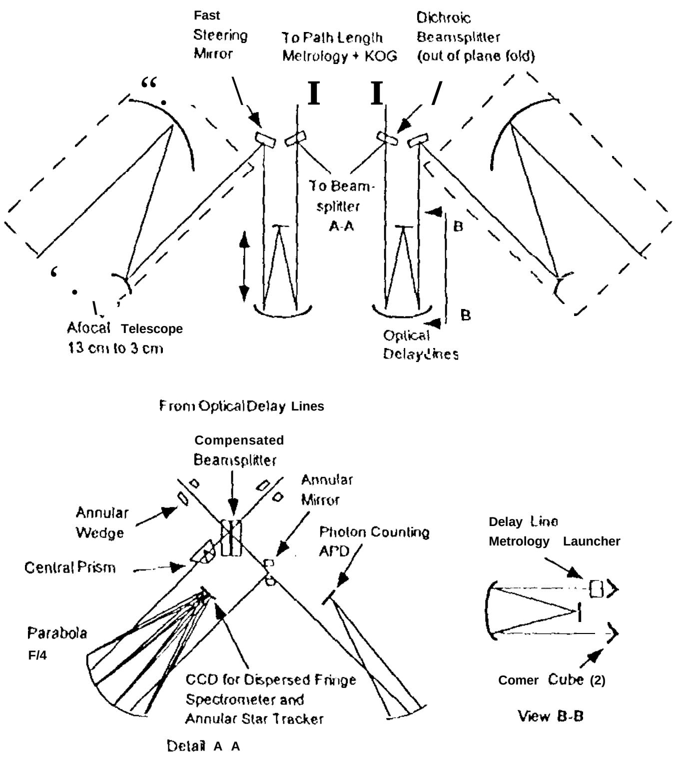

Figure 2: NMI beam combiner concept

2 STARLIGHT SUBSYSTEM

The starlight subsystem begins with the collector gimbals on the two collector spacecraft. These are three-axis gimbals (tip, tilt, and roll), with range to accommodate the assumed ± 0.05 deg. spacecraft attitude deadband. This is consistent with a design approach which seeks to minimize instrument-spacecraft interactions, where fine pointing and phasing control is provided by the instrument, with a dynamic range such that closed-loop spacecraft control is not needed. NMI uses 12-cm clear aperture flat mirrors which direct light to beam-compressor telescopes on the combiner spacecraft; corner-cube retroreflectors are located at the center of the gimbal mirrors for use by the laser metrology system. NMI uses flat collectors for simplicity, allowing uniform array expansion. For a larger system a better approach is for the collector optics to produce a beam waist at the combiner spacecraft, so that only a single large optic is required. This latter approach mandates a fixed collector- combiner separation, unless the curvature of the collector mirrors is made variable.

Figure 2 shows a concept for the beam combiner in the combiner spacecraft; the concept is similar to systems used in ground interferometers. Beam compressor telescopes receive the light from the collector spacecraft and compress it to 3-cm beams, while fast-steering mirrors (FSMs) provide high-bandwidth tilt control. The combination of tilt control by the starlight gimbals and the FSMs maintains the beam overlap and wavefront tilt of the interfering light beams.

Optical delay lines are used in each arm for fine pathlength control. These are implemented as parabola/flat cat's eyes. A range in delay of 2 cm with nanometer resolution accommodates the formation-flying deadband, without the need for nanometer control of spacecraft position. The short delay-line range allows the use of a two-stage (PZT, voice-coil) system, simplifying implementation. Closed-loop pathlength control to less than 10 nm is routinely accomplished with ground versions of similar delay lines.

Detail A-A of Fig. 2 shows a concept for the interferometer beamsplitter and fringe-detection back end. The two outputs of a compensated beamsplitter feed two detectors: a photon-counting APD detector for high sensitivity fringe detection, and a fast-framing CCD. The pupil of the beamsplitter output feeding the CCD is divided spatially between fringe sensing and tilt sensing. The inner part of the beam passes through a direct-view prism) providing a dispersed fringe pattern on one line of the CCD. The annular portion of the beam is]Ion-dispersed, providing images of the two input pupils on the CCD. Annular wedge prisms in each arm offset the annular images from each other and from the dispersed fringe, allowing the same CCD to serve for both tilt sensing and fringe detection. Also shown in Fig. 2 is the injection of the pathlength metrology and KOG, discussed is Sees. 3 and 4, below.

The nominal frame rate for the CCD is 250 Hz, and the combination of APD and CCD detectors allows for fringe detection using temporal modulation accomplished by scanning the optical delay line, or using the dispersed fringe on the CCD with a fixed delay offset. The ultimate sensitivity of the system is estimated as 14 mag, limited by the coherence tirnc provided by the KOG.

3 LASER METROLOGY

The absence of a structure means that structural rigidity is achieved actively, rather than through reliance on the intrinsic stiffness of a metering truss. The sensors for achieving rigidity are the liner metrology systems which measure inter-spacecraft distances. As shown in Fig. 2 laser metrology is introduced into the starlight path through a dichroic beamsplitter in the beam combiner, and measures the distance from the combiner to the corner cubes at the pivots of the starlight gimbals on the two collector spacecraft. The third laser path is implemented separately between the two collector spacecraft; the error budget is such that separate corner cubes (from the ones on the gimbal mirrors) can be used for this measurement, as the star being observed is always very close to perpendicular to the baseline,

Given the pathlengths sensed by these three metrology beams, one could control the positions of the spacecraft to the nanometer level to stabilize the interferometer. It is however more efficient to use the data for feedforward control to the optical delay lines to stabilize the starlight path, correcting for both baseline changes as well as internal pathlength changes. 'l'he delay-line range (2 cm) establishes the required accuracy of the stationkecping of the individual spacecraft. This represents one of several implernentation options, in which there is nominally no feedback between the delay-line control system and the spacecraft control system, and the delay line is sized to accommodate the formation-flying dead band. Alternatively, with tighter systems coupling, the delay-line range could be reduced.

The metrology system would use heterodyne techniques, as used for ground interferometers and proposed for space missions such as SIM, which readily provide <<10 nrn position accuracy. The laser source would nominally be a 1319-nnI diode-pumped single-frequency device, in order to provide a narrow linewidth to maintain coherence over the 2-km maximum round-trip propagation. Heterodyne implementation would use fiber-fed frequency shifters to provide the necessary frequency offset between polarizations.

Finally, two other laser beams are used internal to the combiner spacecraft. These monitor just the delay-line positions, and are used in processing of the KOG signals, discussed below, in order to separate delay-line position changes from spacecraft position changes.

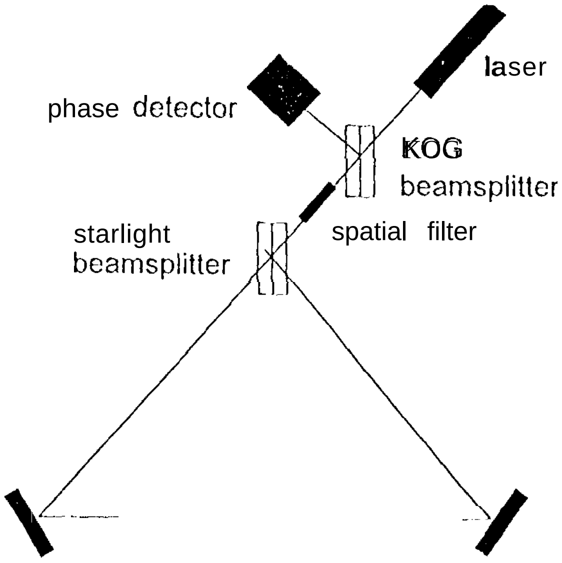

Figure 3: Kilometric Optical *Gyro* concept

4 KILOMETRIC OPTICAL GYRO

Laser metrology pathlength sensing, in conjunction with feedforward compensation by the optical delay lines, provides equivalent structural rigidization. It now remains to point the interferometer constellation so as to not blur the fringe. The conventional approach to this problem uses off-axis stars for sensing. This approach is problematic with long baselines for two reasons. The primary reason is that a small off-axis angle translates into a large delay change: 3 arcmin over a kilometer baseline is 1 meter of path delay, which is not easily accommodated with this architecture; in fact with the simple collector mirrors, the light would miss the combiner entirely. Thus accommodating off-axis stars would require increases in complexity for both the collector and combiner spacecraft, which are presently relatively simple. 'J'he other reason the off-axis guide stars are problematic is simply that long baselines resolve the nearby bright sources which would ordinarily serve as guide stars, reducing the available SNR for tracking.

An alternative approach is to use a purely inertia] sensor. The long baselines of NM I allow for the use of a Kilometric Optical Gyro or KOG, which was proposed for SS1 (Separated-Spacecraft Interferometer proposal to the Code SZ New Mission Concepts NRA). The KOG is a Sagnac interferometer employing counter-propagating beams among the three spacecraft- -essentially a fiber-optic gyro where the fiber sensing coil is replaced by the three-spacecraft propagation- as shown schematically in Fig. 3. This is a particularly good match to the long baselines of a separated-spacecraft interferometer, as the sensitivity of the KOG is proportional to the enclosed area ($\sim$ baseline^2), while the required pointing accuracy scales with baseline, so that the KOG works better with long baselines.

The KOG signal, i.e., the phase shift $\Delta\phi$ between the counter-propagating beams, can be written

$$\Delta\phi = \frac{4\pi\Omega Bh}{\lambda_K c} = \frac{8\pi A\Omega}{\lambda_K c}, \tag{1}$$

where Ω is the rotation rate, B and h are the baseline length and the height of the triangle, A is the area $\frac{1}{2}Bh$, and λ_K is the sensing wavelength. To maintain high fringe visibility, the change in the interferometer delay at science wavelength λ_S must be limited to $\Omega_{\max}Bt = \lambda_S/n, (n \sim 10)$, during a coherent integration time t. Substitution into Eq. 1 yields

$$\Delta\phi = \frac{4\pi\lambda_S h}{n\lambda_K ct}. \tag{2}$$

With $n = 10$, $\lambda_K = 2.5\lambda_S$, and a minimum detectable phase $\Delta\phi_{min}$, the achievable coherence time is

$$t = \frac{h}{2c\Delta\phi_{min}}, \tag{3}$$

which is proportional simply to the height of the triangle. Assuming an equilateral triangle with $\Delta\phi_{min} = 1\ \mu$rad, the achievable coherence time is 0.15 s with B = 100 m and 1.5 s with B = 1 km. This all assumes a perfectly rigid triangle. in reality, the spacecraft are going to be moving. However, by monitoring separately the length of the individual legs of the triangle, it is possible to separate rotation from length change. This is the reason for the separate metrology of the delay lines, described above. While not needed for control reasons, subtraction of this measurement from the combiner- collector metrology, which also passes through the delay lines, gives a measure of just, the inter-spacecraft distance.

The KOG beam is injected through a dichroic beamsplitter in the beam combiner (Fig. 2), and propagates toward each collector. To reflect the KOG light around the loop of spacecraft, rather than back toward the stellar source, a diffraction grating is placed on the collector mirrors. The grating is nominally implemented as a second-surface grating, with the first surface coating a dichroic to reflect starlight and transmit the KOG light, althoughother approaches may allow for a first-surface implementation. Pointing Of the grating is accomplished using the third (roll) axis of starlight gimbal, with sensing using edge- and angle-sensors around the periphery of the gimbals. The KOG would nominally operate at 1.5 μm, distinct from the starlight and laser-metrology wavelengths. 'J'he irnplernentation of the KOG may be able to use a modified fiber-optic gyro sensor head, with changes to the internal electronics to accommodate changing loop pathlengths.

The KOG measures rotation of the constellation as a whole at high frequencies, up to the reciprocal of the KOG coherence time, at which point fringe-trackingon the star in the beam combiner takes over. Similarly accelerometers, opposition sensitive detectors (PSDs) measuring the tilt of the KOG beam, , provide high frequency tilt infornationfor dim sources, with starlight tilt sensing using the beam-combiner CCD detector taking over at lower frequencies.

5 FORMATION FLYING AND INITIALIZATION

As discussed above, the proposed architecture requires formation flying accuracy to 1 crn to avoid saturation of the optical delay lines. However, precision formation flying is only one of several requirements on the propulsion system; others include the implementation of 1) baseline orientation changes, to rotate the instrument about the line-of-sight, sweeping out a chord in the (u,v) plane; 2) baseline length changes, to vary the angular resolution; and 3) retargetting the interferometer to point at other objects. The NM1 point design considered both a small cold-gas system, possible for a short-duration technology -demonstration mission, as well as an electric propulsion system using a pulsed-plasma thruster.

Key to formation flying is a means for initializing the constellation, i.e., sensing the inter-spacecraft relative distances and angles. An innovative sensor concept based on GPS technology (in particular, the JI'L-developed TurboRogue) was developed for this and other applications.[2] The Autonomous Formation Flying sensor (AFF) uses GPS-like signaling among multi-channel tranceivers on the three spacecraft. Each spacecraft thus transmits a phase and pseudorange signal which is received by multiple antennas on the other spacecraft. Multiple patch antennas on each spacecraft allows for both 4rr steradian angular coverage as well as determination of relative angle in addition to range. The target accuracy for the AFF is 1 cm relative distance and 1 arcrnin relative angle, consistent with the formation-flying requirements given above.

Table 1: NMI mission and spacecraft characteristics

Mission	
Mission duration	6 months
Orbit	Heliocentric, (<0.1 AU at 6 months)
Launch vehicle	Delta-Lite
Attitude Control	
Fiber gyro	0.1 deg/hr
Star tracker	10 arcsec, 10 Hz
Pointing control	±0.05 deg
Data Storage	100 MB solid-state memory
AFF	
Relative attitude	±1 arcmin
Relative position	±1 cm
Relative velocity	±0.01 cm/sec
Telecommunications	
Combiner-collector	500 kbps, half duplex, 4 freq.
Combiner-Earth	100 kbps, full duplex, 2 hrs/day
Propulsion	
Control	6 deg. of freedom
Thrust	> 4 mN
Stationkeeping	±1 cm

6 MISSION DESIGN

To minimize the disturbance environment in which the precision formation flying must be achieved, an Earth-trailing heliocentric orbit is proposed. To minimize costs, the mission lifetime is limited to 6 months. A simple observation scenario is proposed, in which the plane of the three spacecraft is nominally perpendicular to the sun vector (to within ~30 deg), providing a uniform thermal environment, simplifying stray-light baffling, and allowing for fixed solar arrays. All stars are observable over the 6-month mission, although the achievable (u,v) coverage will vary.

ROM mass estimates allow for launch in a single Delta-Lite vehicle to the heliocentric orbit. After launch, the individual spacecraft are released from the launch adapter, despun, and the triad configuration formed. The AFF sensor, in conjunction with the individual spacecraft ACS sensors, are employed in this process. The propulsion system is sized to allow for 1000 6-degree *maneuvers* at a 100-m baseline, which was the simplified metric used for fuel calculations.

Two propulsion systems were considered, One design used 12 4.5-mN GN_2 thrusters, with 15 kg of propellant per spacecraft to meet the mission requirements. An alternative design explored electric propulsion implemented with teflon Pulsed Plasma Thrusters (PPTs). At a 6 Hz pulse rate, the thruster considered (Olin Aerospace) provides ~4.2 mN thrust, while the higher I_{sp} allows for only ~1 kg total propellant per spacecraft.

'J'able 1 summarizes some of the spacecraft and mission characteristics resulting from a point-design study at JPL.

7 CONCLUSION

The NM] concept is a simplified separated spacecraft interferometer with the goal of technology demonstration to enable future applications of interferometer and other spacecraft constellations. Key technologies include those of space interferometry in general: starlight fringe detection, laser metrology, and active optics. Technologies specific to NMI include the Kilometric Optical Gyro for inertial phasing of the constellation, and the Autonomous Formation Flying sensor for illter-spacecraft ranging. Testbeds for demonstration of key technologies and for systems integration and test will also be required for successful implementation of NM],

8 ACKNOWLEDGEMENTS

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